STARS MDT-II TARGETS MISSION

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The Strategic Target System (STARS) was launched successfully on August 31, 1996 from the Kauai Test Facility (KTF) at the Pacific Missile Range Facility (PMRF). The STARS II booster delivered a payload complement of 26 vehicles atop a post boost vehicle. These targets were designed and the mission planning was achieved to provide a dedicated mission for view by the Midcourse Space Experiment (MSX) Satellite Sensor Suite. Along with the MSX satellite, other corollary sensors were involved. Included in these were the Airborne Surveillance Test Bed (AST) aircraft, the Cobra Judy sea based radar platform, Kwajalein Missile Range (KMR), and the Kiernan Reentry Measurements Site (KREMS). The launch was a huge success from all aspects. The STARS booster flew a perfect mission from hardware, software and mission planning respects. The payload complement achieved its desired goals. All sensors (space, air, ship, and ground) attained excellent coverage and data recording.

Introduction

The Strategic Target System (STARS) Mission 3/MSX Dedicated Target mission (MDT-II) was successfully launched from the Kauai Test Facility (KTF) on the Pacific Missile Range Facility (PMRF) on Kauai, Hawaii at 15:41:49 GMT on August 31, 1996 (see figure 1). This mission was conducted under the Ballistic Missile Defense Organization's (BMDO) Consolidated Targets Program. The United States Army Space and Strategic Defense Command (USASSDC) is the executing agent for this mission and the Department of Energy's (DOE) Sandia National Laboratories (SNL) is the vehicle developer and integrator. This target mission included the deployment of 25 test objects from an Operational Deployment Experiment Simulator (ODES) post-boost-vehicle. These test objects were viewed by the Midcourse Space Experiment (MSX) satellite, and a variety of air, sea, and ground based auxiliary sensors. Impact occurred about 216 nm northeast of the Kiernan Reentry Measurement System (KREMS) complex on the Kwajalein Missile Range (KMR). Participating auxiliary sensor platforms included the Airborne Surveillance Testbed (AST) aircraft, the Cobra Judy (USNS Observation Island) radar ship, and a variety of sensors on the KMR. The STARS booster and post-boost vehicle operation were nominal, and data in support of mission objectives was collected by all participating sensors.

This paper begins with a brief description of the MSX satellite. Later sections will detail the target mission scenario, describe the MDT-II STARS booster configuration, explain the pre-flight hardware and software testing, and provide a summary of the flight test results.

MSX Satellite [1]

The MSX satellite is the first in-space system demonstration of technology to characterize ballistic

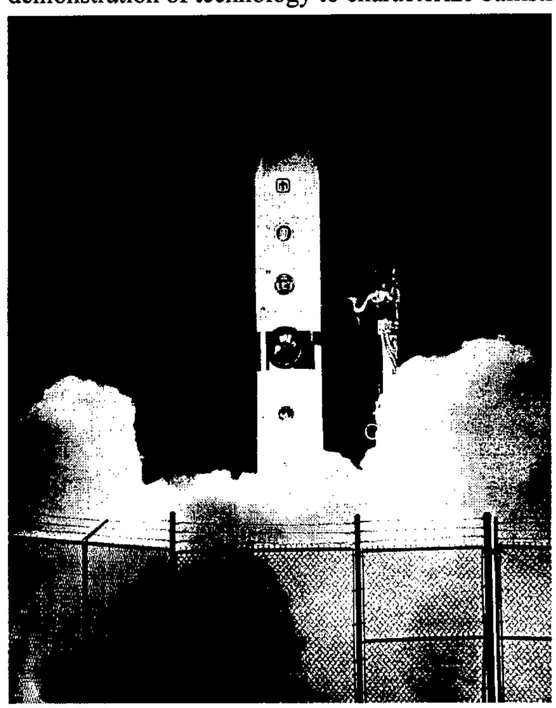

Figure 1. STARS M3/MDT-II Launch

missile signatures during the midcourse flight phase between boost and missile reentry. The satellite is designed to detect, track, and discriminate realistic targets against terrestrial, Earth limb, and celestial

backgrounds. MSX is capable of observations over a wide range of wavelengths, from the very-long infared to the far-ultraviolet.

The spacecraft has five primary instruments consisting of 11 optical sensors. All sensors are precisely aligned so that simultaneous observations with multiple sensors can be made, which is essential for scenes or targets which change rapidly. These primary instruments include the SPIRIT III (Spatial Infared Imaging Telescope), UVISI (Ultraviolet Visible Imagers and Spectrographic Imagers), SBV (Space Based Visible), OSDP (On-Board Signal and Data Processor), and several contamination sensors. The SPIRIT III was built by the Space Dynamics Laboratory of Utah State University. The UVISI and the contamination sensors were built by the Johns Hopkins University Applied Physics Laboratory. The SBV sensor was built by the Massachusetts Institute of Technology Lincoln Laboratory, and the OSDP was built by Hughes Aircraft Company.

The MSX satellite was launched on April 24, 1996 from Vandenberg Air Force Base, into a high-inclination, circular, near sun-synchronous orbit at 487.9 nm (903.5 km) altitude. The satellite was built and operated by the Johns Hopkins University Applied Physics Laboratory.

MDT-II Target Mission Plan

Planning for the STARS M3/MDT-II targets mission was driven by a complicated set of requirements related to viewing by the MSX satellite on-board sensors, as well as by ground, sea, and air-based supplementary sensors. The MDT-II targets mission included requirements for functional demonstrations of metric and radiometric discrimination, Post Boost Vehicle (PBV) and deployed object acquisition and tracking, as well as cluster track and resolution. Other requirements included collection of target signature phenomenology on the STARS booster and PBV, deployed reentry vehicle replicas and penaids, and clusters of objects. Other practical considerations also resulted in mission constraints. The targets mission launch window was required to be five days long to accommodate weather or technical delays, and to reduce range scheduling complications. This requirement allowed for up to five launch attempts per window, since the launch opportunity on each of these days was only a few seconds long. All of the mission objectives were incorporated into the definition of a "feasibility" region, which would ensure satisfaction of mission requirements if the satellite groundtrack was located within this "feasible" geographic region at the time of target launch. The feasibility region was directly related to the target trajectory and deployment timeline, and several iterations on each were used to help ensure the satisfaction of mission objectives.

Another complication in the planning process was the data collection requirements of the auxiliary sensors. Generally, these issues were handled by the placement of the mobile sensor platforms, and via compromises in the target trajectory and mission timeline, particularly in the case of the fixed, ground-based assets at KMR.

Basic target trajectory requirements included a minimum apogee of 3,000,000 ft to ensure space background tracking during the post-boost phase of the mission, and a maximum range from the KREMS complex of about 300 nm to provide best data from ground-based assets on the KMR. All target trajectories were designed to ensure that the 3-sigma low performance case satisfied the minimum apogee altitude requirement.

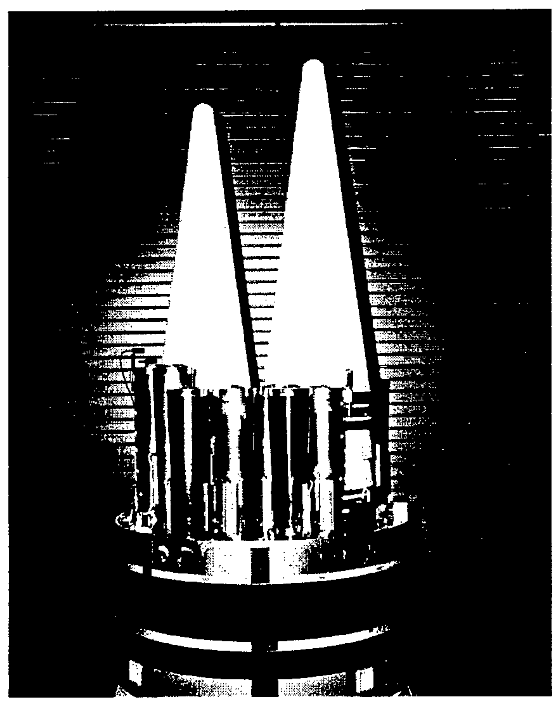

Figure 2. MDT-II Targets and PBV

Off-nominal performance during the STARS boost phase was one determinant of the feasibility region boundary. None of the STARS motors is equipped with a thrust cut-off capability (except for the flight

termination system, which would be used only to ensure range safety), and off-nominal performance has a direct bearing on the resulting target trajectory. Several third stage targeting options were evaluated during the mission planning process to produce a target trajectory with the least restriction on the feasibility region. Two basic third stage guidance options are available with the STARS system. Both are based on solutions to Lambert's problem, which provides for a target point constraint. The first option is an energy wasting procedure designed to constrain time-of-flight. In this case all trajectories resemble the 3-sigma low case, with significant maneuvering during third stage burn, and nearly constant time-of-flight. The second option provides for a nearly constant attitude third stage burn, a variable time-of-flight, and no energy wasting. In this case the resulting trajectory shape will vary with booster performance, but the desired target point will be achieved at a variable time. Target points off of the 3-sigma low trajectory baseline at pre-apogee, apogee, post-apogee, and impact were evaluated. After extensive analysis by the mission planning team, the second guidance option was chosen with a pre-apogee target point. This scenario provided for acceptable trajectory variations during the pre-apogee deployment phase, while directing all available energy into delivering the targets near to the KREMS complex on the KMR.

The MDT-II mission started with first stage ignition initiated at a time determined daily based on the latest orbit data of the MSX satellite. The launch window was only a few seconds long on any given day, so a fixed time associated with the middle of the feasibility region was chosen as the launch time for that day. Any problems encountered late in the countdown generally required a slip to the next day. The first stage initial fly-out azimuth was also set late in the countdown, based on launch day winds, to ensure adequate range safety margins as the instantaneous impact point (IIP) traversed north of the inhabited island of Niihau west of Kauai. The launch day selection of launch azimuth was required due to the transition to open-loop guidance after the initial pitch-over was completed at 20 seconds into the mission. At this time an angle-of-attack control logic was enabled which attempted to fly zero angle-of-attack through the staging event. This controller used inputs of launch day winds from balloon flights to fly near true zero angle-of-attack so as to improve control margins for the extended length STARS II booster. Closed-loop guidance was reestablished 10 seconds after 2nd stage ignition, and the turn towards the KMR was initiated. After second stage burn-out, a minimum separation coast of 10 seconds was employed prior to third stage ignition. The short coast period provided for the earliest possible third stage ignition, which resulted in the best overall trajectory range and earliest payload deployment times. During this coast the spent 2nd stage was retroed away, the initial guidance solution for the third stage was completed, and an attitude control system (ACS) reorientation maneuver was executed to setup for the third stage burn. Approximately 5 seconds into the coast phase the clamshell shroud separation was initiated. The ACS system was disabled just prior to and then re-enabled just after the shroud separation event. At the end of the 10 second coast period, the third stage motor was ignited. After a nearly constant attitude burn, the PBV and payload complex were essentially on the required path to the target point. At third stage burn-out a 12 second coast was initiated to prepare the PBV for the separation event. During this time, the ACS was active to maintain control of the vehicle as the third stage tailed-off. Once the PBV fuel and electrical systems were activated, the separation of the PBV and third stage was initiated. The PBV propulsion system was used to provide a separation velocity, and was also used to eliminate any off-nominal performance of the third stage. After a series of maneuvers, the PBV was situated to begin payload deployments. At the end of the payload deployment sequence, several PBV maneuvers and engine burns were executed for viewing by the MSX satellite. A final satellite experiment was conducted late in the mission as the PBV engines were turned-on again to be viewed with an earth limb and upper-atmosphere interactions.

A variety of targets were deployed during the MDT-II mission. All targets were designed and built by SNL. There were 26 test objects to be deployed during the MDT-II mission, of which 10 were instrumented with the SNL Light Weight Instrumentation System (LWIS). These included an instrumented MSX Aeroshell (MAS), 2 ultra-light replicas (ULR), 6 Rigid Light Replicas (RLR) (3 instrumented), 7 Canisterized Light Replicas (CLR) (4 instrumented), 4 Canisterized Traffic Balloons (CTB) (2 instrumented), an Emissive Reference Sphere (ERS), and a Multi-Balloon Canister (MBC). Telemetry data from the instrumented targets was relayed to the ground via a re-rad system onboard the PBV. A SNL designed encrypted video system was also located on the PBV to provide real-time confirmation of test object deployment, and reference data for post-flight analysis. The integrated PBV and payload is shown in figure 2.

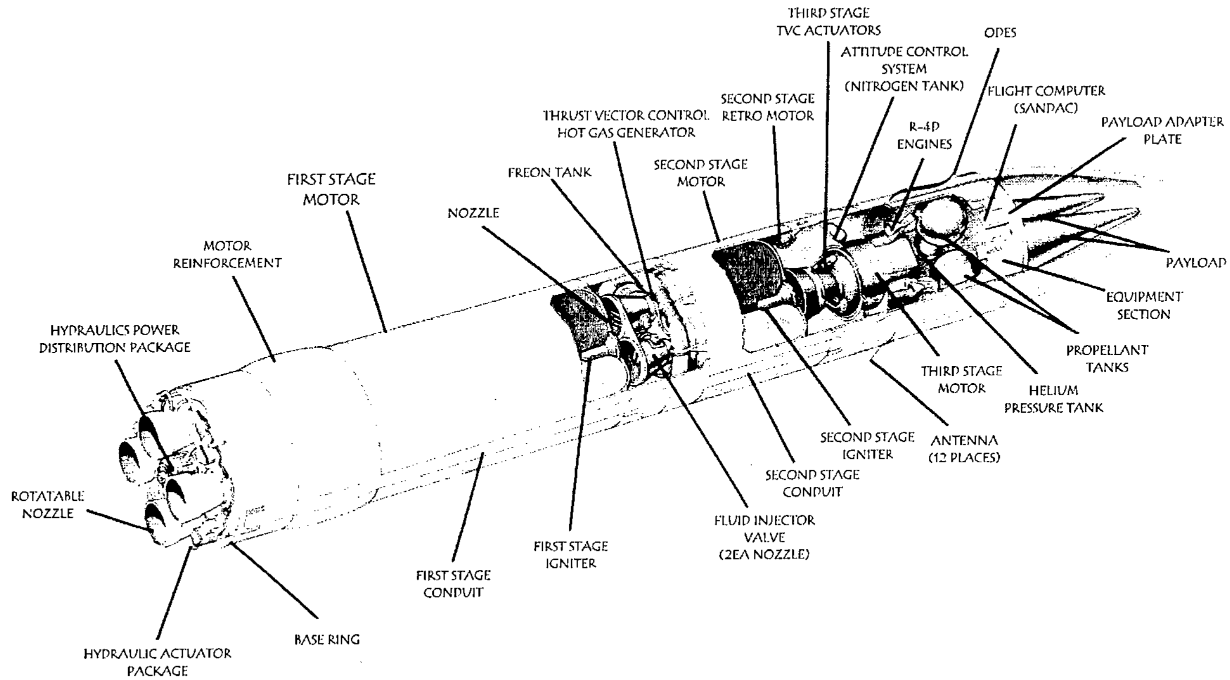

Figure 3. STARS II Booster Cutaway

Because of the quantity of targets involved, and the requirement by auxiliary sensors to switch from object to object in real time during the mission, GPS corrected IMU state vectors were generated onboard the STARS PBV and telemetered to the KTF. Subsequently, these state vectors were transmitted to the auxiliary sensors to be used as aids to improve real-time radar acquisition within the complicated target complex. This capability has now been successfully demonstrated on the last two STARS missions and provides increased reliability in achieving mission data objectives for complicated target missions.

STARS II Booster [2]

The STARS II is a three stage solid rocket motor booster with a liquid fueled PBV called the Operational Deployment Experiment Simulator (ODES). This vehicle is shown in figure 3.

The STARS II first stage (FS) rocket motor is a refurbished Polaris A3P FS motor. The stage is 182 inches (15.2 feet) long, 54 inches in diameter, and contains approximately 20,800 pounds of solid propellant (high explosive equivalency of 10,500 1bs TNT). The STARS II FS motor assembly consists of an externally modified motor casing, an igniter assembly, four rotatable nozzle assemblies with hydraulic actuation for thrust vector control (TVC), and onboard flight termination system (FTS). The motor chamber is a fiberglass filament wound pressure vessel with skirts and bosses, filled with solid propellant. An external casing modification has been made to the motor case that consists of a 24 inch-wide composite overwrap, centered 31 inches forward of the aft skirt. The composite overwrap is a hot gas seal consisting of a casing bonded insulation blanket under fiberglass hoop wraps. Motors that are selected for flight must pass a full body radiographic inspection. The igniter assembly consists of a pyrotechnic igniter chamber and a new 28 volt hot bridge-wire initiator. Four equally spaced rotatable nozzle assemblies are bolted to the nozzle bosses in the motor aft dome. The nozzles are controlled by individual hydraulic actuator packages (HAPs) that are powered by a hydraulic power distribution package (HPDP) centered on the aft dome. Prior to flight, the nozzles are removed from the candidate flight motor, recertified by leak and torque testing, and reinstalled. The HAPs and HPDPs are certified for flight by acceptance testing of the individual components and functional testing of the system.

The FS FTS is located on the forward dome of the motor. The FTS consists of flexible linear shape charge (FLSC) bonded to the dome, two MC3644

detonators with attachment hardware, and the MA170 FTS electronics package with auto-destruct capability. The motor thrust can be terminated by cutting the fiberglass forward dome with the FLSC and venting the internal motor pressure.

The STARS II interstage section (IS) is a modified Polaris A3 IS. It is a cylindrical shell made of a magnesium-thorium sheet, HK31A-H24, which is 54 inches in diameter and 34.3 inches long. The first stage motor ignition umbilical interface is on the IS. A mild detonating fuse (MDF) is installed around the forward circumference of the section to separate the first stage and interstage from the rest of the missile after first stage burnout. The MDF is initiated by two MC3644 detonators. The interstage section contains an A3 Hydraulics Battery for powering the FS TVC. The original A3 interlocks have been removed.

The STARS II second stage (SS) rocket motor is a refurbished Polaris A3P SS motor. The stage is 89 inches (7.4 feet) long, 54 inches in diameter, and contains approximately 8,800 pounds of solid propellant (high explosive equivalency of 8712 lbs TNT). The STARS II SS motor assembly consists of a modified motor chamber, an igniter assembly, four fixed nozzle assemblies with liquid injection for TVC, and an onboard FTS. The motor chamber is a fiberglass filament-wound pressure vessel with skirts and bosses that is filled with solid propellant. The chamber modification is internal in the forward dome region. The refurbishment modifications involve draining the liquefied potting out of the gap between the chamber insulation and the propellant shrinkage liner; repotting the gap with a silicone material; and replacing the rigid potting containment device with a flexible potting containment baggie. The baggie collects residual liquefied potting and prevents the potting from contacting the propellant. The igniter assembly consists of an igniter chamber and a new 28 volt hot bridge-wire initiator. The main charge of propellant in the igniter has the basic chemical properties of the SS motor propellant. This new initiator also ignites the hot gas generator (HGG) in the SS TVC system. Four equally spaced, fixed nozzles are attached to the nozzle ports by retaining rings. The flight motor candidates are received and x-rayed without nozzles installed. The nozzles are inspected separately, refurbished (if required), and certified for flight prior to installation. The SS liquid TVC system consists of four injector valve assemblies, a manifold assembly, a toroidal fluid filled tank, a HGG, and a hot gas relief valve. Second stage thrust deflections are accomplished by injecting fluid into one side of a fixed nozzle to create a shock wave. Nozzles 1 and 3 are used for pitch control while nozzles 2 and 4 are used for yaw control. Roll control is accomplished using all nozzles. All TVC components are certified for flight by inspection and acceptance testing. The manifold and injectors are functionally tested with nitrogen during assembly and final systems checkout.

The SS FTS is located on the forward dome of the motor. The FTS consists of FLSC bonded to the dome, two MC3644 detonators with attachment hardware, and the MA170 FTS electronics package with auto-destruct capability. The motor thrust can be terminated by cutting the fiberglass forward dome with the FLSC and venting the internal motor pressure.

The STARS II third stage (TS) assembly is a new design replacing the old Polaris A3 equipment section (ES). The TS consists of a structure which houses an Orbus 1 motor and an interstage between the second and third stages. The new TS is 62.8 inches long and 54 inches in diameter. The skin consists of a 0.160 inch magnesium-aluminum alloy sheet, AZ31B-H24. A 0.030 inch thick coating of Dow Corning 92-009 applied to the exterior surface of the TS protects the magnesium skin from aerodynamic heating during the ascent through the atmosphere.

The TS section houses the STARS II downstage missile electronics. The forward end of this section is defined by the mounting ring for adaption of the post boost vehicle (PBV), and the aft end by the lower MDF separation system. The four missile umbilical interfaces, including the payload systems umbilical, are located here. The electronic systems housed in this section include; 1st 2nd and 3rd stage TVC packages, attitude control system (ACS) with pneumatics, and the third stage FTS auto destruct package. The Orbus 1 motor is a new motor with TVC designed for STARS to SNL specifications by United Technologies Corporation, Chemical Systems Division (UTC/CSD). The motor is 49.2 inches long and 27.2 inches in diameter. It contains approximately 910 pounds of UTP-19687A solid propellant (high explosive equivalency of 920 lbs TNT) with an HTPB binder. The Orbus 1 motor assembly consists of a graphite composite case, an igniter assembly, a nozzle assembly with a flexseal joint, an electro-mechanical TVC system with thermal battery, and an onboard FTS. The igniter assembly is a toroidal pyrogen igniter initiated by dual Teledyne McCormick Selph electric low voltage detonators, part number 817447. During third stage

burn, vehicle pitch and yaw are controlled with the motor TVC system. Roll attitude is controlled during burn using the cold gas ACS. The Orbus 1 FTS is located on the aft dome of the motor. The FTS consists of FLSC bonded to the dome and two MC3644 detonators with attachment hardware. The motor thrust can be terminated by cutting the graphite aft dome with the FLSC to separate it along with the nozzle assembly from the rest of the motor.

The ODES PBV is mounted to the adapter ring on the front of the TS. The PBV is separated from this structure in flight by a debris free MDF separation joint just below the adapter ring. The PBV is constructed with a composite monocoque design utilizing balsa wood, carbon fiber and aluminum. This design allows for high strength with a lightweight structure. The PBV is divided into two discernible sections; the Component Section (CS) and the Propulsion Section (PS). The PS contains all hypergolic fuel tanks and fuel delivery system along with FTS related hardware. The PS also contains the R4D rocket motors, their actuator mounting systems, and the electro-mechanical actuators (EMA). The CS section contains all of the flight electronics, which is required for fulfilling the mission requirements after all stages are separated. These systems include the Guidance Navigation and Control System (GN&C) which includes; Ring Laser Gyro Assembly (RLGA) inertial measurement unit (IMU), Sandia Digital Airborne Computer (SANDAC), GPS receiver and supporting equipment. Also, included are the PBV TVC packages, Arm and Fire (A&F) system with programmable sequencer, telemetry support hardware, antennas and all system batteries.

The MDT-II STARS II utilized a newly designed composite clamshell shroud, which was jointly developed by McDonnell Douglas and Sandia National Laboratories [3]. This shroud split into two halves upon separation. The shroud provided a RF transparent cylindrical section to allow all PBV mounted antennas to radiate through the shroud. The shroud also contained a GPS antenna in the nose tip for pad and early ascent GPS viewing. The fully stacked missile with shroud is 38.6 feet tall which is 5 feet longer than the STARS I configuration.

Development and Testing

Once the MDT-II payload delivery requirements were determined, those which drove modifications to the existing STARS II design were recognized. As typical with most launch vehicles, reducing weight and increasing performance were critical.

Design and Development

The existing system, as flown on the STARS M2/ODES Development Flight (ODF), was evaluated and two major mechanical modifications were required. First, the PBV was targeted for weight reduction primarily through lightening of the structure and payload plate. Also, all components which could reasonably be modified for weight reduction were repackaged and in some cases redesigned. Second, the existing one piece STARS shroud ejection method was too slow to meet requirements; therefore, a clamshell shroud was designed to reduce the coast time between 2nd stage burn out and 3rd stage ignition. Performance analysis indicated that igniting the third stage earlier than scheduled in the standard STARS timeline gained approximately 1.4 pounds of payload mass to the target for each second of reduction in coast time. A 40 second reduction in coast time, which was enabled by the use of the clamshell shroud, allowed the growth of the payload by about 60 pounds, the single most significant performance gain of all the performance modifications.

Based on previous lessons learned, MDT-II requirements, and weight reduction issues, a number of components were targeted for redesign or repackaging. The A&F sequencer was redesigned for capacitive discharge. This design tends to limit impact on flight batteries during ordnance firing operations, and adds significant channel capability, up to 108 firing lines. The PBV TVC control electronics were redesigned based on weight reduction and improved servo control functionality. The SANDAC flight computer was upgraded from Motorola 68020 to Motorola 68040 processors. This modification reduced the number of processors required and achieved weight reduction. The system junction box was also redesigned and packaged for a significant reduction in weight. Each new or repackaged component was passed through a qualification test which is more strenuous than the typical flight acceptance testing which is required of all flight components before being deemed flight worthy. Both qualification and acceptance include shock, vibration, and temperature cycling. Some components also require resonant bar (pyro shock) and vacuum testing.

Once the components were designed and connector's defined, mock ups were fabricated and each component (actual or mock) was mounted physically on the structure (PBV or 3rd stage). This placement was determined either by mass properties or

functional dependence. After these components were placed and the pinouts and connectivity requirements known, the cable routing and lengths were determined. The design allowed for component replacement and reduction of conductors in a given cable. This reduced cost and impact for redesign of cables for problems found during testing. There are about 180 cables in the PBV, and about another 50 on the remaining portions of the missile. These cables go through pin to pin testing, hipot testing and in all cases go through system level environmental testing while components are functioning.

Modifications to components and mission requirements drove changes to ground support equipment and ground test simulators. The Ground Launch Computer (GLC), telemetry decoms, missile simulators and other mission specific or programmable equipment were modified and checked out. Once the support equipment function was validated, it was deemed to be safe to connect flight hardware to these systems.

Test & Evaluation (Albuquerque)

The acceptance tested components were mounted into the PBV and cable connections made. Initial electrical continuity checks were performed and then a comprehensive procedure was performed to connect the integrated system to tested ground support equipment. At this point the system had the ability to be powered up and down as prescribed by the GLC software or as required for testing. The 3^{rd} stage components were also installed and the cable connections made. A similar procedure for checkout was also run on this part of the vehicle.

The PBV was mounted on the system test stand, which is a controllable moving base which provides independent roll, pitch and yaw motion for IMU input. This input drives the GN&C system in the current flight sequencer mode and tests the hardware-in-the-loop system with live input and response. Lower stages were then electrically connected, one by one, to test their functionality to these inputs.

First, the PBV autopilot test was run. The GN&C and PBV TVC system were activated and put into the correct autopilot mode. Tests were performed to confirm that the motion was passed correctly through the inertial sensors, software paths, and processed correctly throughout the autopilot code. The actuators were also tested independently for range-of motion, accuracy, and bandwidth.

The attitude control system test followed (see figure 4.). This test was different from the other control system tests since the system actually controls the vehicle through a single axis maneuver (roll, pitch and yaw are all tested). The stacked PBV and 3^{rd} stage (with mass mock 3^{rd} stage motor) were hung below a crane hoisted air bearing for relatively frictionless rotational motion about the vertical axis. The handling fixture allowed the vehicle to be put in all cardinal orientations. This test again confirmed correct attitude motion was passed through sensors and software. Also, the attitude control code was exercised and control loop gains were confirmed (based on test moments of inertia). Finally, solenoid action and the pressure delivery system functions were tested and confirmed.

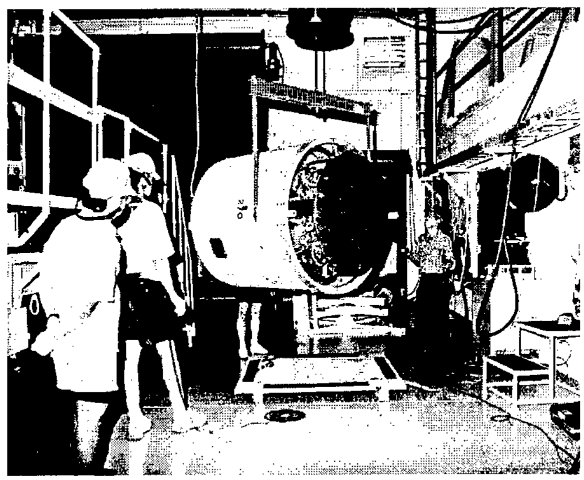

Figure 4. Air Bearing Testing of the ACS

Next, the 3^{rd} stage autopilot was tested. The setup for this test was basically the same as the PBV autopilot except that extension umbilical cables were connected from the base of the PBV to the 3^{rd} stage structure. Also, a Heavy Weight System Test Motor (HSTM), an actual Orbus 1 motor with inert motor grain, was loaded into the 3^{rd} stage structure. The correct GN&C mode was selected and the motion is converted into correct pitch and yaw actuator deflections and roll ACS commands with nozzle firings. Actuator performance was also determined during this sequence of testing to evaluate TVC flight acceptance procedures.

The 2^{nd} stage autopilot was tested next. This system was configured with the PBV on the system test stand with umbilical connections to the 3^{rd} stage and the 2^{nd} stage. This test produced servo motion of the fluid injectors with gaseous nitrogen at about 180 psi. Functionality was demonstrated by this test, however,

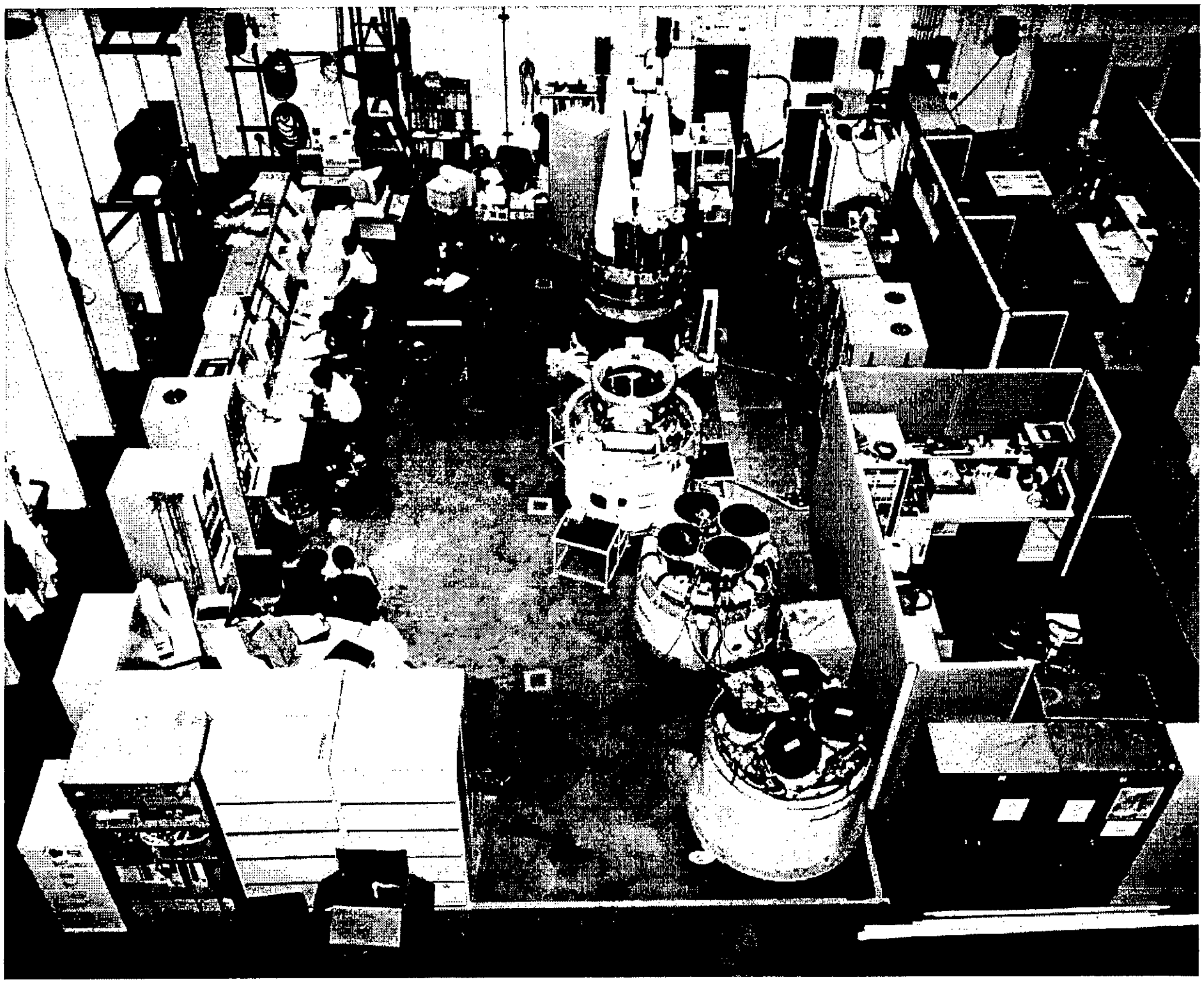

Figure 5. STARS M3/MDT-II Vehicle Testing in the Albuquerque Laboratory

servo loop gains are not tested, since the flight system utilizes liquid freon at about 750 psi rather than nitrogen. However, the test did confirm that all flight hardware/software and telemetry hardware was functioning correctly.

The 1st stage autopilot was then tested. This system continued the extension cable hookup from 2nd stage to 1st stage. This tested the gains of the 1st stage autopilot code, servo loops, and hydraulic actuators of the unloaded 1st stage nozzles. The actual flight environment operates under a considerably different nozzle load due to engine thrust passing through the nozzles. The actuator bandwidth is evaluated and compared to data derived from loaded actuator tests.

A preliminary system test (hardware-in-the-loop flight sequence checkout) was then conducted (see figure 5.). This test was the culmination of all the testing so far. The system was configured with the PBV on the test stand with all stages electrically connected. Next, the flight sequence was confirmed by placing bridge wire simulators in all connectors to electro-explosive devices. The test was run by the GLC, the onboard flight computer and A&F sequencer. The GLC sent correct navigator modes and the initial launch discrete to start the on board sequence. The GLC also initiated motion of the system test stand at prescribed times in the flight sequence to provide IMU input to test TVC and ACS modes and functionality. The corresponding bridge wire simulator firings at the correct times confirmed all systems were functional and working per mission requirements.

At this point the missile system was prepared for payload integration.

The payload interfaces were completely checked for continuity and electrical signals. The payload interface hardware was then mounted to the payload plate on the PBV. The payloads were temporarily mounted next. Note that this was not the final payload integration. The payloads were then checked out through the use of payload groundstation hardware and powered by the GLC. Once all payloads were checked, the full system was ready for testing.

Another system test was run, which effectively was the same as the previous except that the payload sequencing and firing were also checked per the mission timeline and requirements. This concluded the pre-environment system testing of the integrated vehicle.

Independent of the Albuquerque testing, the first & second stage motors were refurbished at their respective manufacturers. The 1st stage motor was then delivered to China Lake NAWC for a computed tomography (CT) scan. Similarly, Alliant Tech Systems conducts a Linatron scan on the 2nd stage motor. Components for the TVC systems of the 1st and 2nd stage were certified by a number of methods and shipped to Hill AFB for integration. Once the motors were delivered to Hill AFB for assembly, the thrust vector control (TVC) system actuators and related TVC and FTS components were mounted and preliminary checkouts conducted. This was the final test of the flight motors prior to shipment to the Kauai Test Facility (KTF).

Environmental Testing

The stacked 3rd stage, PBV and payload were then installed onto the shaker at the Sandia Environmental Test Facility for vibration and shock testing. This system was configured to independently test roll, pitch, and yaw axes. Each axis was tested at varying levels of the vibration and shaker shock specification. The system is functioned electrically through these environments.

The pyro-shock test of a mock 3rd stage, PBV and payload was then performed. This test stacked the PBV and payload onto the mock 3rd stage with an actual 3rd stage-PBV separation joint installed. The system was functioned electrically during the test. The test fired the separation joint and actually cut the metal that transmitted a flight like shock through the system.

Once the environmental testing was completed and the system was deemed flight worthy, a final system test in Albuquerque was run. This test effectively repeated the test prior to environments to confirm all systems were still in working order.

Software Validation

Flight software validation was required to provide quality control, test and evaluation of the flight code. In some cases this environment provided the only flight like test of the software. The flight software was functioned in an identical SANDAC flight computer with a flight like A&F sequencer, PCM encoder, GPS receiver, 1553 bus and other components which were then hooked into the simulation computer through a custom interface. This computer simulated the missile dynamics, TVC actuators, and other system models. The software was then executed in a significant number of parametric runs to test robustness of flight control and other flight sequence software. The software was certified as flight ready when all parametric runs were completed with the final version of code.

After the hardware was deemed flight worthy and ready to ship to the launch facility a Vehicle Readiness Review (VRR) was held. This meeting presents vehicle status, anomalies, and waivers. At that time all participants evaluated the status and determined readiness to commit to the field. The missile, payload, and all support equipment were then shipped to KTF/PMRF.

Testing and Integration (Kauai)

The entire shipment of flight motors, ordnance, booster hardware, payload and support equipment were shipped via Military Airlift Command (MAC) to the Pacific Missile Range Facility (PMRF). The day of shipment all items were offloaded and put into the respective buildings for assembly, test and storage at the Kauai Test Facility (KTF) (see figure 6.).

Initial Checkout and Preparation

Basically, the Albuquerque checkout procedures were performed again to confirm all systems were ready for launch.

The PBV and 3rd stage were inspected to confirm all components and cables were mounted and connected per procedure. Also, serial number and component

identification were recorded in the Missile Build Book for completeness and acceptance confirmation.

The flight Inertial Measurement Unit (IMU) was gyrocompassed and evaluated independent of the flight vehicle in the GN&C lab of the Missile Assembly Building (MAB). This testing was performed on a surveyed granite table and evaluated prior to installation into the PBV.

Figure 6. The Kauai Test Facility (KTF)

In parallel, the MAB ground support equipment was checked out and any required modifications were made prior to connection to the flight vehicle.

Also, the Launch Operations Building (LOB) systems were configured for the flight vehicle. The missile personnel configured the STARS racks, which contain the GLC, Booster TM, FTS, GN&C and support PC's, and control consoles. These consoles communicate and control operation of the missile and support hardware in the MAB and Missile Service Tower (MST)/Auxiliary Equipment Building (AEB). The payload personnel configured payload TM, control and battery charging hardware, and other support equipment in their respective positions in the LOB. The KTF personnel continued with range support and configuration, which included FTS transmitter, TM receiving and distribution along with all range support functions.

Testing & Evaluation (Kauai)

The PBV EMA's were calibrated on a test stand with independent measurements corresponding to the data acquired by the flight computer. This formed the calibration utilized by the flight software. Next, the PBV was mounted on the system test stand and the PBV autopilot procedure was performed as in Albuquerque.

Next, the ACS testing was performed. The PBV was mounted to the 3rd stage structure (which had a 3rd stage mass mockup installed). And the ACS system was prepared by filling the high pressure bottles with nitrogen to the prescribed pressure. This testing confirmed the polarity of the nozzles (solenoids) and the operability of the system. Then the integrated vehicle (PBV and 3rd stage) was hung from the crane hoisted air bearing and single axis maneuvers were performed. This confirmed solenoid delays, control loop gains and basic functionality of the complete system, including power, TM, control electronics, battery operation, etc.

The 3rd stage motor, Orbus 1, was prepared and installed into the 3rd stage structure. Thermal insulation was checked on the underside of the 3rd stage structure and around the base of the Orbus motor. The PBV was then remated to the system test stand. Umbilicals were connected from the PBV to the 3rd stage and the system was ready for 3rd stage autopilot testing. This was the first time the flight electronics were connected to the flight motors. From that point forward, all powered tests were run remotely from the LOB, due to the hazardous nature of powering electronics in close proximity to explosives. Since the test was remote, video cameras with fiber optic feeds to the LOB were utilized for visual verification of TVC polarity along with telemetry.

The 2nd stage TVC and autopilot testing was then performed. This test required umbilicals from the 3rd stage to the 2nd (since the booster had not yet been stacked). Custom pressure instrumentation was installed in each nozzle and connected to a strip chart recorder. This provided independent verification of the nozzle TVC polarity along with the TM data. The instrumentation was then removed and the 2nd stage autopilot procedure was performed remotely.

The 1st stage autopilot test was performed next. The 1st and 2nd stages were temporarily mated, completing the full hook up of the system downstage (all TVC's connected). Video cameras were set up looking back at the nozzles of the 1st stage. This video provided the independent verification of the actuator polarity along with TM.

With the system completely connected electrically, a preliminary system test was run to confirm flight and

ground hardware and software were working correctly. Next, the payload interface hardware and payloads were installed and the system was readied for the full up system test. A dry run system test was run to confirm all systems were functioning correctly and all personnel were aware of their respective duties. The bridgewire simulators were loaded and readied for testing. The system test procedure was then executed in remote mode, and all flight systems were verified for functionality with the flight timeline.

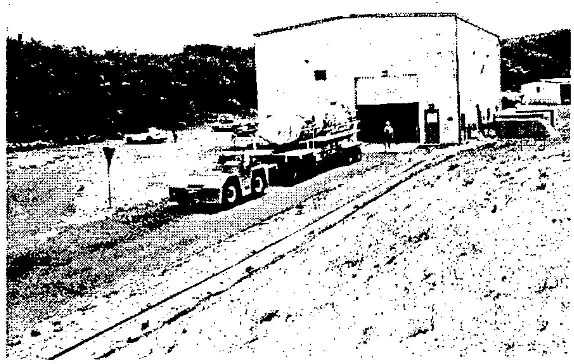

Figure 7. Missile leaving the MAB

Missile Final Integration

All remaining ordnance items were installed at this time, prior to final integration. This included mild detonating fuze (MDF) and initiators for skin separations, initiators for motor ingiters, etc. These operations were performed with a minimum number of authorized personnel. During this period, no other operations were performed in the MAB.

With the ordnance installed, the 1st and 2nd stages were aligned and mated per procedure on the transporter erector. The 3rd stage and PBV were then soft mated for the missile assembly test. This was a system by system state of health check to ensure that all connections were properly configured.

Next, the PBV was removed and the STARS lifting shroud was installed. The booster was then stored in the MAB, ready to be moved to the pad and erected on the launch stool.

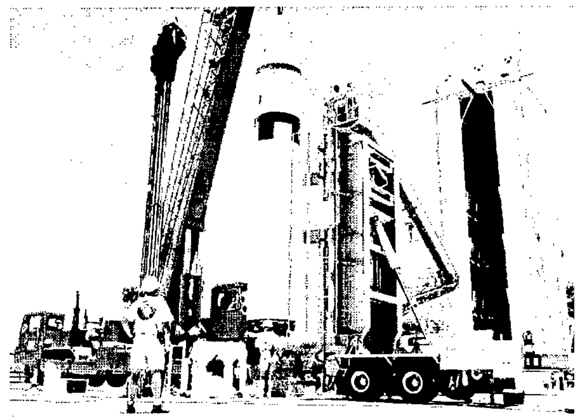

Figure 8. Missile being erected on pad.

Payload integration was then performed. The PBV was moved to an assembly stand, which was better suited for complete access to the payload plate. The payload personnel then installed all flight payloads with ordnance for the final time. At the completion of this stage the integrated PBV and payload were ready to be moved to the launch pad (MST) for hypergolic fueling of the PBV.

Pad Operation Procedures

The integrated PBV and payload were moved to the launch pad and installed under the fueling tent which was connected to the MST. Fueling operations were performed per procedure with a minimum number of personnel present due to the hazardous nature of the propellant and oxidizer. Once the fueling was completed, the fueled PBV was moved to the environmentally controlled shelter connected to the AEB. The fueling tent was then taken down and the pad area cleared.

The MST was removed to the retracted position and the pad area prepared for missile erection. The integrated 1st/2nd/3rd stage with the lifting shroud was moved to the pad on the transporter/erector (see figure 7.). This vehicle was moved to a prealigned location along with a crane and man lift. A coordinated lift was performed once the missile was erected with the transporter/erector. The crane was then attached to the lifting shroud. The missile, once free of the erector, was placed on the launch stool within a degree of the desired roll angle orientation (see figure 8.). The MST was then placed back around the missile for environmental control and to provide access to work levels.

With the missile erected and the MST in place, the lifting shroud was removed, exposing the top of the 3rd stage. The integrated and fueled PBV with payload was then lifted with handling gear, utilizing the MST crane. The PBV was then mated (umbilicals and in-flight disconnects) and all attach hardware installed.

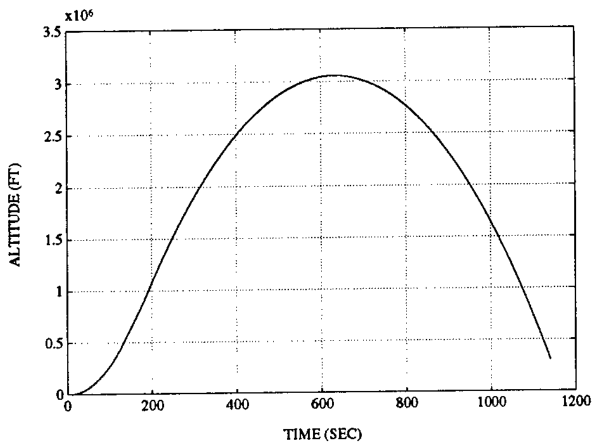

Figure 9. Altitude Versus Time

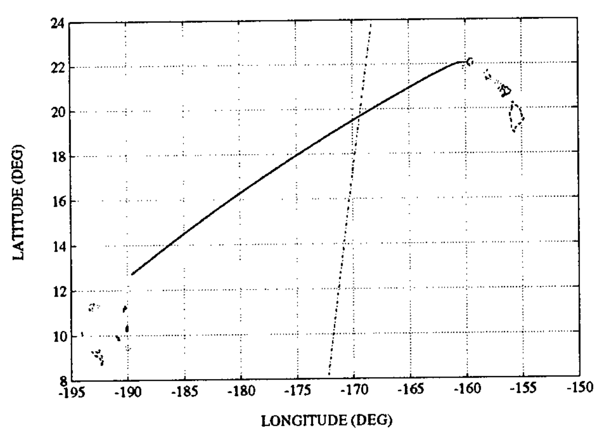

Figure 10. Ground Track

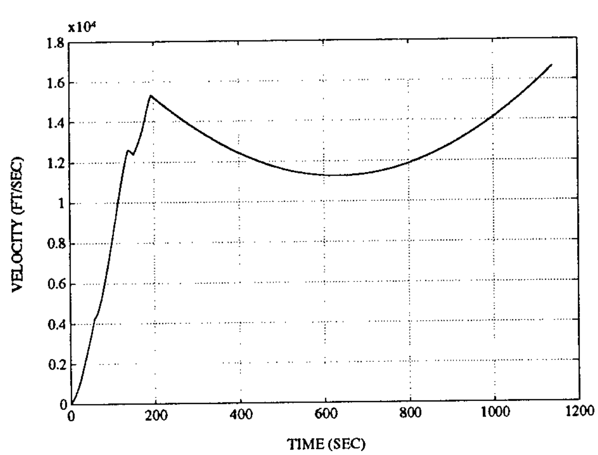

Figure 11. Total Velocity Versus Time

With the system fully integrated and all umbilicals attached, the missile was ready for post mate integration testing. The tests exercised the state of health of all systems and determined that the vehicle was ready for flight. This process spanned a number of days.

Once flight worthiness was determined to be satisfactory, shroud installation and final ordnance system connections were completed. The Flight Readiness Review (FRR) was conducted at this time. This meeting included all participants and provided a forum to present the status of all flight hardware and any anomalies. Each system was discussed, along with any deviations from procedure. The outcome of this meeting was a GO for final launch preparations. The system was then ready to begin dry run count downs.

Count Down Preparation

All personnel at KTF involved in the actual launch began practice counts and preparations for flight. These counts tied all missile, payload, KTF range systems, and personnel to the common goal of the safe and reliable launch of the missile in the required launch window. These dry runs highlighted rough areas which required procedural modifications and also gave personnel practice to provide good confidence that all operations could be performed on time to meet the desired launch window.

The next step was to perform count downs which included the PMRF personnel, hardware, and software. This provided the same function as the previous dry runs but checked out more required systems and personnel.

Finally, the Mission Readiness Test (MRT) was run. This was a dry run with all participants on the clock, including KTF, PMRF, KMR, corollary sensors, and others. All pre launch procedures were executed just like the actual launch. The countdown proceeded through T=0 and simulated flight data was passed on to all ranges and sensors as if the launch had actually occurred.

Prior to flight, a Mission Readiness Review (MRR) was held. The MRR was a comprehensive review of the MRT. Based on the successful MRT and the GO/NO GO status poll of the mission participants, the BMDO Program Manager authorized the launch of the STARS M3/MDT-II mission.

Flight Test Results

STARS Booster Performance

A simplified MDT-II flight timeline is shown in Table 1. This timeline shows most of the major events of the target mission. Most of the timing variations relative to nominal preflight predictions were due to burn time variations in the boost phase rocket motors. Mission sequencing was designed to ensure that test object deployment times were fixed relative to launch to facilitate sensor acquisition. All deployed objects were released at the prescribed nominal times, with the exception of test object #17 (MAS), which did not deploy due to a failure in the payload customer supplied V-band release mechanism. Apogee and reentry times were somewhat earlier (9 and 15 seconds respectively) than nominal prediction due to a slightly low boost phase performance.

The boost phase flight trajectory is shown in Figures 9-10. Figure 9 shows altitude versus time, while Figure 10 shows the STARS groundtrack for the entire mission, as well as the MSX satellite groundtrack (dashed line) for mission times between 582 and 858 seconds. The trajectory apogee was about 25,000 ft. less than the nominal prediction, but well above the 3,000,000 ft. minimum requirement. The third stage vacuum impact point was about 24 nm short of the nominal preflight prediction. Figure 11 is a plot of total velocity versus time. Total velocity was approximately 95 ft/s lower than predicted. These variations in the trajectory were the result of a lower than nominal net boost phase performance, but were well within the bounds established for mission success. Overall boost phase performance was within 1-sigma of nominal. Target miss for the third stage through the desired pre-apogee point was about 0.30 nm.

Performance variations for each of the individual stages can be observed from acceleration data. Thrust acceleration vs. time plots are shown in Figures 12-14 for stages 1-3, respectively. These figures show flight data (solid line) compared with the preflight predictions (dashed line). First stage

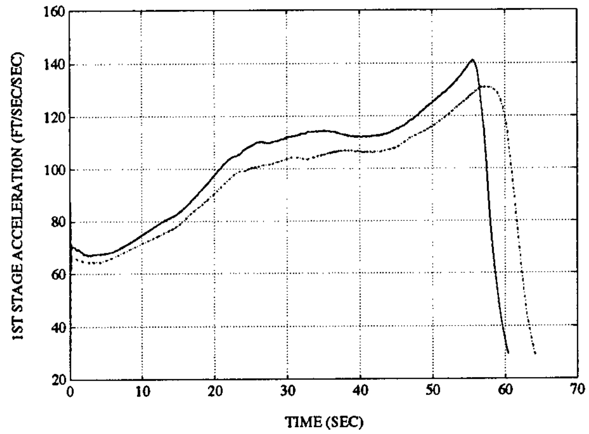

Figure 12. First Stage Acceleration

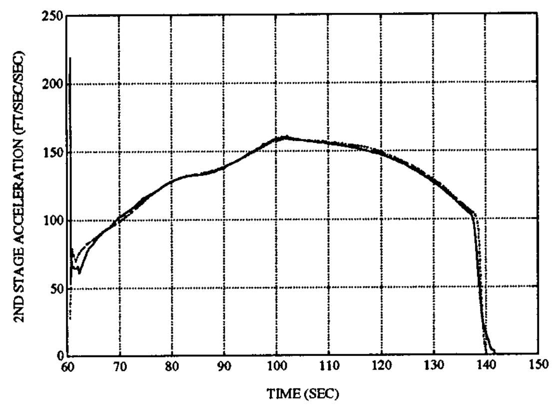

Figure 13. Second Stage Acceleration

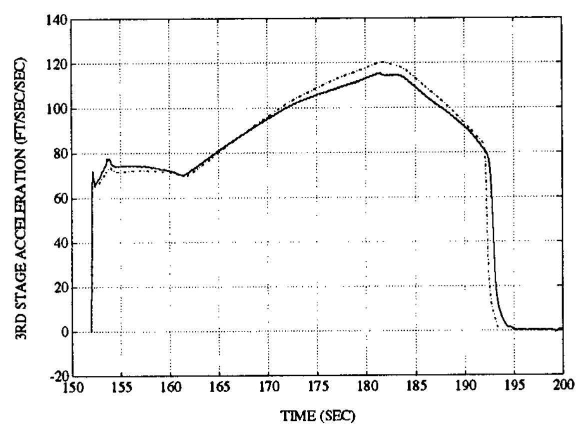

Figure 14. Third Stage Acceleration

integrated acceleration was about 1.5% lower than

preflight predictions, but the burn time was 3.7 seconds shorter than nominal. The second stage integrated acceleration was approximately 0.7% low with a burn time that was 1.5 seconds long. Third stage integrated acceleration was only 0.1% low with a burn time that was 1.2 seconds longer than predicted. Each of these individual variations was within the expected performance parameters for the respective motor.

The ODES PBV performed as expected. PBV propulsion temperatures and pressures were nominal, as well as fuel usage.

MISSION EVENT	TIME	MISSION EVENT	TIME
IGNITION COMMAND	0.0	DEPLOY TO13	371.5
BEGIN PITCH-OVER	2.2	DEPLOY TO14	373.5
BEGIN AOA CONTROL	20.0	DEPLOY TO15	376.0
1ST/2ND STAGING	60.5	DEPLOY TO16	378.0
BEGIN NIIHAU TURN	70.5	DEPLOY TO17	(380.0)
2ND/3RD STAGING	141.6	DEPLOY TO18	382.0
ACS MANEUVER	142.0	DEPLOY TO19	383.5
SHROUD JETTISON	147.0	DEPLOY TO20	385.0
3RD STAGE IGNITION	152.0	DEPLOY TO21	386.5
3RD STAGE BURN-OUT	194.5	DEPLOY TO22-TO26	396.5
PBV SEPARATION	206.5	TURN FOR BURN #1	459.0
TURN TO DEPLOY	250.0	BEGIN BURN #1	475.0
WAIT TO DEPLOY	268.0	END BURN #1	488.0
DEPLOY TO1	300.0	TURN FOR BURN #2	503.0
DEPLOY TO2	302.0	BEGIN BURN #2	515.0
DEPLOY TO3	304.0	END BURN #2	530.0
DEPLOY TO4	306.0	TURN FOR BURN #3	545.0
DEPLOY TO5	307.5	BEGIN BURN #3	557.0
DEPLOY TO6	309.0	END BURN #3	572.0
DEPLOY TO7	311.0	TURN FOR BURN #4	592.0
DEPLOY TO8	312.0	BEGIN BURN #4	618.5
DEPLOY TO9	313.0	END BURN #4	625.0
DEPLOY TO10	314.0	BEGIN COAST	625.0
DEPLOY TO11	364.0	TURN FOR REENTRY	1082.7
DEPLOY TO12	369.5	BEGIN BURN SEQUENCE	1102.7

Table 1. MDT-II Mission Timeline

<u>**Mission Data Objectives [4]**</u>

Much data was collected during the MDT-II target mission. Data collection objectives included:
(1) Acquire, track, and observe the ODES PBV and all deployed objects in sunlight
(2) Observe deployed objects with high signal-to-noise in sunlight with SPIRIT III radiometer sensor in selected IR bands
(3) Obtain precision tracking on ODES and deployed objects in support of functional demonstration of metric discrimination
(4) Observe unresolved target clusters as they evolve into separated individual point source images associated with each test object
(5) Test the efficacy of various deployment related techniques
(6) Observe individual deployed objects with SPIRIT III radiometer during and after solar terminator crossing
(7) Observe ODES vehicle with engines on/off at various aspect angles during and after target deployment
(8) Observe STARS II booster and upper stages during powered flight with an earth limb background and after burnout in medium and long wavelength IR
(9) Observe individual targets and penaids as they begin to reenter the atmosphere
(10) Observe the emissive reference sphere to aid in calibration of the SPIRIT III

MSX satellite data collection was outstanding and nearly all (greater than 97%) of the program objectives were achieved. Data acquisition by auxiliary sensors was also excellent. AST observed about 900 seconds of the mission and recorded IR signature data as well as tracking data. Cobra Judy acquired several hundred seconds of track time on the target complex with its X-band radar. Data reduction is in progress, and is likely to require years of detailed analysis.

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[2] "Strategic Target System (STARS), STARS I Handbook for Payload Designers", SM-PH-01, Revision A, Sandia National Laboratories, April 1994.

[3] "Development of the STARS Composite Payload Fairing", M. J. Robinson and M. R. Weber, presented at the 42nd International SAMPE Symposium and Exhibition, Anaheim, CA, May 5-8, 1997.

[4] "Strategic Target System (STARS M-3)/MSX Dedicated Target Mission (MDT-II) Quick Look Report", U.S. Army Space And Strategic Defense Command (USASSDC), Sept. 1, 1996.

Sandia is a multiprogram laboratory operated by Sandia Corporation, a Lockheed Martin Company, for the United States Department of Energy under Contract DE-ACO4-94AL85000.